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TECHNICAL NOTE
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EFFECT OF STREAMLINE CONTOURING IN THE WING-FUSELAGE JUNCTURE IN COMBINATION WITH THE SUPERSONIC AREA RULE ON A SWEPTBACK-WING—FUSELAGE CONFIGURATION OF HIGH FINENESS RATIO

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NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

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EFFECT OF STREAMLINE CONTOURING IN THE WING-FUSELAGE JUNCTURE IN COMBINATION WITH THE SUPERSONIC AREA RULE ON A SWEPTBACK-WING—FUSELAGE CONFIGURATION OF HIGH FINENESS RATIO

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SUMMARY

An investigation has been conducted in the Langley transonic blowdown tunnel at Mach numbers from about 0.67 to about 1.32 to compare two methods for reducing the pressure drag of a high-fineness-ratio sweptback-wing—body combination at a Mach number of 1.3. One configuration was axisymmetrically indented according to the principles of the supersonic area rule and the other configuration employed the streamline concept of shaping the wing-fuselage juncture with the same longitudinal area development as that of the supersonic-area-rule configuration. The ratio of the fuselage frontal area to the total plan-form wing area was 0.0606.

The results of the investigation indicate that, for a relatively slender configuration, the streamline concept of shaping the fuselage does not offer pressure-drag reduction greater than that obtainable with the supersonic-area-rule concept.

INTRODUCTION

A series of investigations have been undertaken to evaluate the concept of shaping the fuselage of a sweptback-wing—fuselage combination in such a way as to combine the curvature of the streamline over an infinite sweptback wing with the longitudinal area distribution obtained from application of either the transonic area rule (ref. 1) or the supersonic area rule (ref. 2). Experimental data are presented in references 3, 4, and 5 which show that this method of shaping the fuselage resulted in reductions in pressure drag coefficient significantly greater than those obtained through the use of axisymmetric application of either the transonic or supersonic area rules alone. In these references, it is believed that the success of the streamline contouring method in reducing pressure drag depends to a large extent upon how well the fuselage aerodynamically separates the two swept-wing panels. An increase in overall slenderness of the configuration or a decrease in fuselage size relative to wing size may result, therefore, in a decrease in the effectiveness of the streamline-contouring method.

The purpose of the present investigation, therefore, was to obtain such experimental data on two sweptback-wing—fuselage combinations of high fineness ratio. Tests were made on one wing-body combination contoured by axisymmetric application of the supersonic area rule alone and on another wing-body combination contoured by the streamline concept combined with the longitudinal area distribution obtained by axisymmetric application of the supersonic area rule.

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The tests were conducted in the Langley transonic blowdown tunnel through a range of Mach numbers from 0.67 to 1.32 with a corresponding variation in Reynolds number from 1.26×10^6 to 1.56×10^6 based on wing mean aerodynamic chord. The angle of attack was varied from about -1.4° to 8.5°.

SYMBOLS

A	aspect ratio
C_D	drag coefficient, $\dfrac{\text{Drag}}{q_\infty S}$
C_{D_0}	drag coefficient at zero lift
C_L	lift coefficient, $\dfrac{\text{Lift}}{q_\infty S}$
C_m	pitching-moment coefficient, $\dfrac{\text{Pitching moment about } \bar{c}/4}{q_\infty S\bar{c}}$
$(L/D)_{max}$	maximum lift-drag ratio
$\bar{c}$	wing mean aerodynamic chord
c	local wing chord
d	diameter

q_∞	free-stream dynamic pressure
M_∞	free-stream Mach number
S	total wing area
α	angle of attack, deg

MODELS

Two fuselages of fineness ratio 10.0 were tested in combination with a wing of aspect ratio 4, taper ratio 0.6, 45° sweepback of the quarter-chord line, and NACA 65A006 airfoil sections in the stream direction. The wing was mounted on the fuselage in the midwing position at an angle of incidence of 0° and had no twist and no dihedral. The ratio of fuselage frontal area to total wing plan-form area was 0.0606.

One of the fuselages tested was axisymmetrically indented according to the principle of the supersonic area rule. The indentation, accomplished in the same manner as reference 2, was designed for a free-stream Mach number of 1.3. Ordinates obtained for the area-rule fuselage are presented in table I and a sketch and photographs of the configuration are presented as figures 1 and 2, respectively.

A second fuselage was obtained by combining the streamline concept of shaping the fuselage with the longitudinal area development resulting from application of the supersonic area rule. The fuselage contour was shaped to conform to the calculated streamline shape that would exist on the wing surface at a free-stream Mach number of about 1.3 if the fuselage was not present and the sweptback wings were of infinite span. The method of calculating the streamline shape for the present investigation was the same as that employed in references 3 and 4. Ordinates obtained for the streamline contoured fuselage are presented in table II and a sketch and photographs of the model are presented in figures 1 and 3, respectively.

The longitudinal cross-sectional area development of the configurations tested and that of a basic wing-body combination are shown in figure 4.

APPARATUS

The tests were made in the Langley transonic blowdown tunnel which has a slotted octagonal test section measuring 26 inches between flats.

The model was mounted on an internal three-component electrical strain-gage balance which was sting-supported in the tunnel. Force and moment data were recorded by self-balancing potentiometers on pen-type strip charts. The base pressure and the pressures necessary to determine dynamic pressure and Mach number were recorded with quick-response flight-type recorders.

TESTS

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The tests were made through a range of Mach numbers from 0.67 to 1.32 with a corresponding variation in Reynolds number from 1.26×10^6 to 1.56×10^6 based on the wing mean aerodynamic chord. The angle-of-attack range was varied from about -1.4° to 8.5°, and the measured angles were corrected for sting and balance deflections due to aerodynamic loads.

All the tests of the present investigation were run with fixed transition in order to avoid changes in aerodynamic forces due to changes in the extent of laminar flow on the models. The roughness strips were constructed of 0.001- to 0.002-inch-diameter carborundum grains blown on a thin layer of wet shellac. The strip on each fuselage was 1/4 inch wide and the leading edge of each strip was located at 10 percent of the fuselage length. Care was taken to assure that the same degree of roughness was applied to each fuselage forebody. The strips on the wings were about 1/8 inch wide and the leading edge of the strips were located at about 10 percent of the local chord from the wing leading edge on both the upper and lower wing surfaces. The wing roughness strips were the same for both configurations inasmuch as the same wing was used for all the tests.

Based on previous experience with models of a similar size, it appears that, for the fuselage alone, the results will be influenced by tunnel wall reflections through a range of Mach numbers between about 1.04 and 1.14. The interference range, however, would be extended to a slightly higher Mach number for the wing-body combination due to reflected disturbances intersecting the wingtips. The wingtip disturbance should have no effect on a comparison of results between the two wing-body combinations inasmuch as the disturbance should not only be small but the same for both configurations.

The drag data have been adjusted to a condition of free-stream static pressure at the base of the model. In addition, the drag data measured at Mach numbers of about 1.16 and greater were corrected for buoyancy effects resulting from gradients in test section Mach number.

RESULTS AND DISCUSSION

The measured lift and pitching-moment coefficients are presented in figures 5 and 6. Presented in figure 7 are the drag polars for both configurations. The following table presents the zero-lift drag coefficients and maximum lift-drag ratios obtained from the tests of the supersonic-area-rule and streamline-contoured configurations at two test Mach numbers that bracket the design Mach number of 1.30.

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Configuration	M_∞	C_{D_0}	$(L/D)_{max}$
Supersonic area rule	1.28	0.0248	5.61
	1.32	0.0251	5.52
Streamline contour	1.28	0.0258	5.56
	1.32	0.0261	5.36

References 3, 4, and 5 indicate that pressure-drag reductions greater than those obtained by axisymmetric application of the principles of the transonic or supersonic area rule have been obtained with configurations having the same longitudinal area distribution and having the fuselage locally shaped according to the natural streamline flow that would exist over an infinite sweptback wing. These references also state that the pressure-drag reduction may decrease as the overall slenderness of the configuration increases or as fuselage size relative to wing size decreases. That this is so is shown by the lack of pressure-drag reduction due to streamline contouring attainable with the present configuration as compared with the reduction attained in reference 4 on a configuration also designed for a low supersonic Mach number. The fuselage fineness ratio of the present configuration was 10 and the ratio of the fuselage frontal area to wing area was 0.0606 as compared with values of 9 and 0.11, respectively, for the configuration of reference 4. It appears then that detailed differences in contouring are not as important for wing-body combinations that approach a theoretically slender configuration.

CONCLUSION

An investigation has been conducted in the Langley transonic blowdown tunnel at Mach numbers from about 0.67 to about 1.32 to compare two methods for reducing the pressure drag of a high-fineness-ratio sweptback-wing—body combination at a Mach number of 1.3. The results of the investigation indicate that, for a relatively slender configuration, the streamline concept of shaping the fuselage does not offer pressure-drag reduction greater than that obtainable with the supersonic-area-rule concept.

Langley Research Center,
National Aeronautics and Space Administration,
Langley Field, Va., February 19, 1960.

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1. Whitcomb, Richard T.: A Study of the Zero-Lift Drag-Rise Characteristics of Wing-Body Combinations Near the Speed of Sound. NACA Rep. 1273, 1956. (Supersedes NACA RM L52H08.)

2. Whitcomb, Richard T., and Fischetti, Thomas L.: Development of a Supersonic Area Rule and an Application to the Design of a Wing-Body Combination Having High Lift-to-Drag Ratios. NACA RM L53H31a, 1953.

3. Howell, Robert R., and Braslow, Albert L.: An Experimental Study of a Method of Designing the Sweptback-Wing—Fuselage Juncture for Reducing the Drag at Transonic Speeds. NACA RM L54L31a, 1955.

4. Howell, Robert R.: Experimental Study of a Method of Designing the Sweptback-Wing—Fuselage Juncture To Reduce the Drag at Moderate Supersonic Speeds. NACA RM L55H05a, 1956.

5. Palmer, William E., Howell, Robert R., and Braslow, Albert L.: Transonic Investigation at Lifting Conditions of Streamline Contouring in the Sweptback-Wing—Fuselage Juncture in Combination With the Transonic Area Rule. NACA RM L56D11a, 1956.

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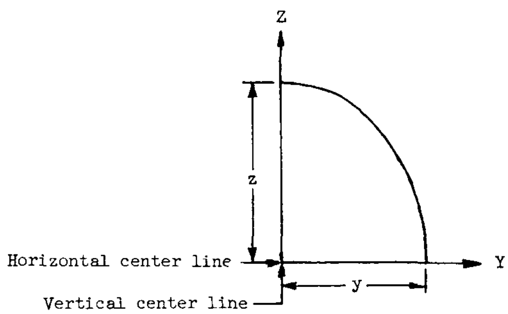
TABLE I.- DESIGN COORDINATES FOR THE FUSELAGE INDENTED ACCORDING TO SUPERSONIC AREA RULE

Body station, x	Body radius, r
0	0
.25	.122
.50	.172
.75	.211
1.00	.243
1.50	.298
2.00	.345
2.50	.385
3.00	.422
3.50	.456
4.00	.487
4.50	.490
4.65	.483
5.15	.437
5.65	.397
6.15	.400
6.65	.422
7.15	.433
7.65	.439
8.15	.438
8.65	.431
9.15	.419
9.65	.404
10.00	.400

TABLE II.- DESIGN COORDINATES FOR THE COMBINATION FUSELAGE

Body station, x	Body radius, r
0.00	0.000
.25	.122
.50	.172
.75	.211
1.00	.243
1.50	.298
2.00	.345
2.50	.385
3.00	.422
3.50	.456
4.00	.487
4.50	.500
4.65	See body cross-sectional ordinates
4.85	
5.15	
5.65	
6.15	
6.65	
7.15	
7.65	
8.15	.438
8.65	.424
9.15	.418
9.65	.404
10.00	.400

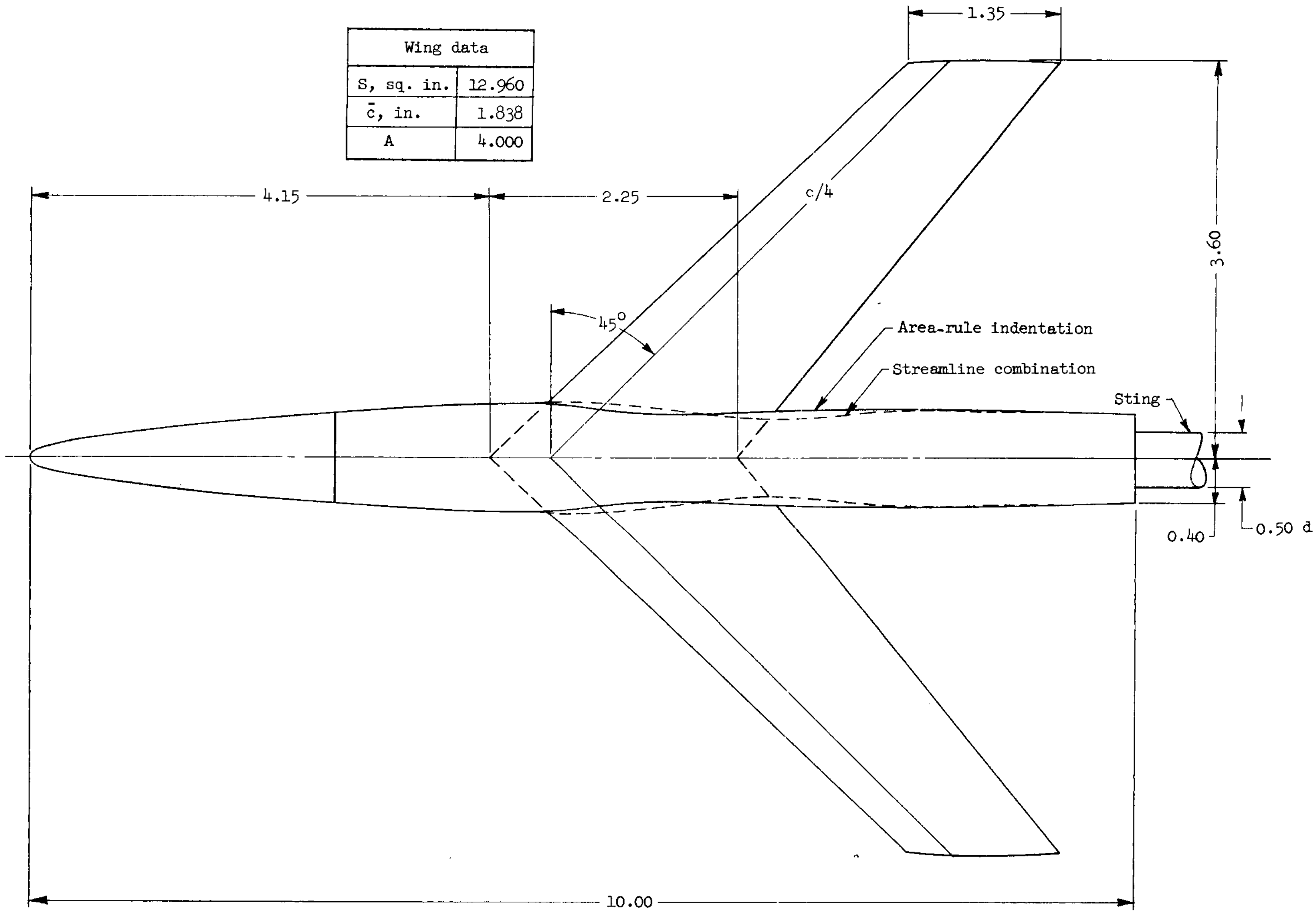


Body cross-sectional view

Body cross-section ordinates

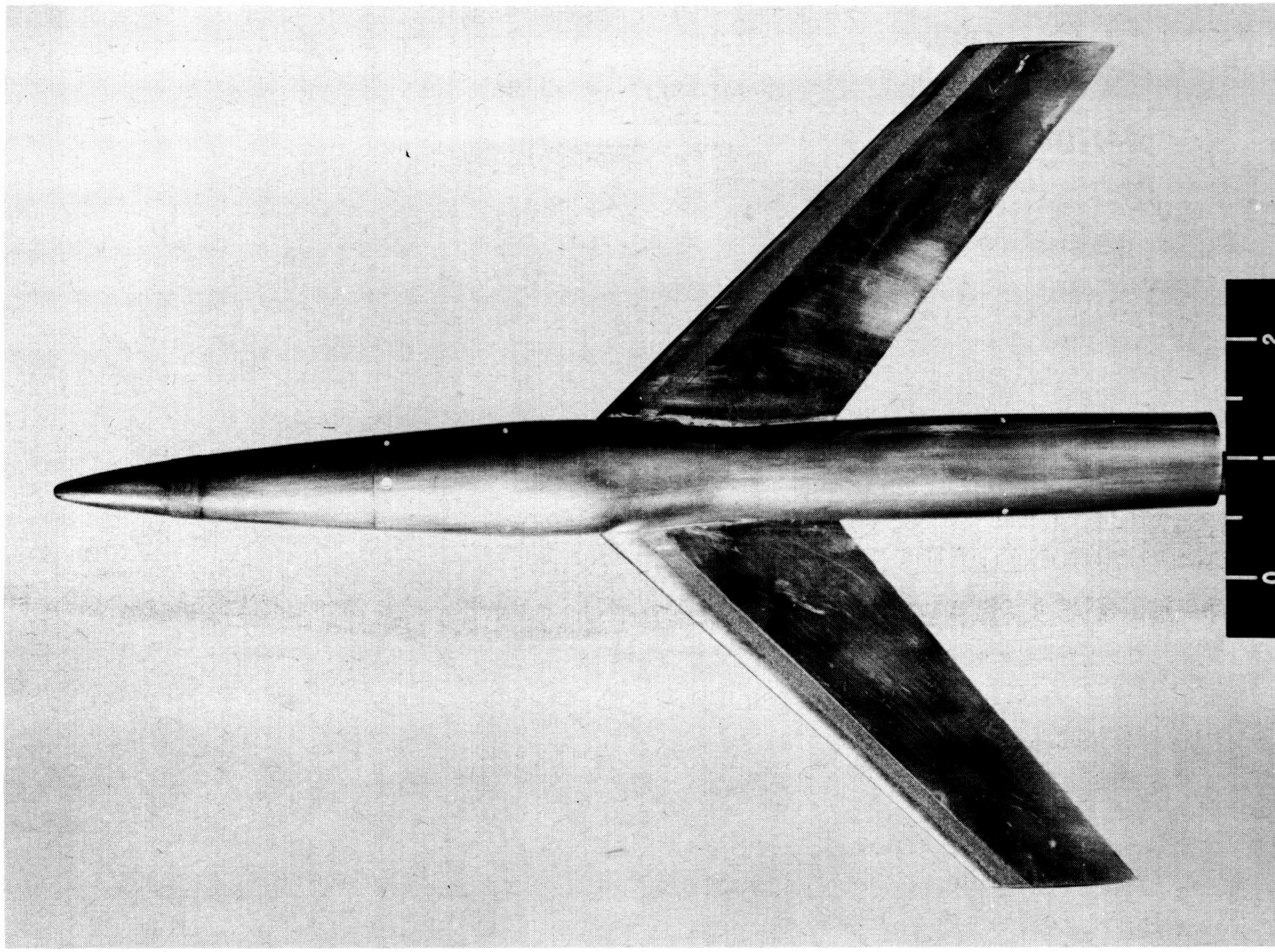
x = 4.65		x = 4.85		x = 5.15		x = 5.65		x = 6.15	
z	y	z	y	z	y	z	y	z	y
0.500	0	0.507	0	0.494	0	0.450	0	0.387	0
.497	.050	.504	.050	.490	.050	.445	.050	.384	.050
.489	.100	.493	.100	.478	.100	.432	.100	.376	.100
.453	.200	.476	.150	.461	.151	.408	.150	.361	.150
.424	.250	.450	.200	.425	.200	.372	.200	.339	.200
.385	.300	.414	.250	.380	.250	.320	.250	.309	.250
.334	.350	.367	.300	.318	.300	.241	.300	.267	.300
.265	.404	.300	.350	.223	.350	.166	.330	.208	.350
.176	.440	.197	.400	.121	.380	0	.355	.103	.400
0	.470	0	.434	0	.392			0	.415

x = 6.65		x = 7.15		x = 7.65	
z	y	z	y	z	y
0.347	0	0.368	0	0.415	0
.345	.050	.367	.050	.413	.050
.341	.100	.361	.100	.406	.100
.332	.150	.352	.150	.393	.150
.320	.200	.338	.200	.375	.200
.303	.250	.321	.250	.350	.250
.282	.300	.298	.300	.317	.300
.255	.350	.268	.350	.273	.350
.219	.400	.230	.400	.212	.400
.170	.454	.176	.450	.134	.440
0	.515	0	.512	0	.465



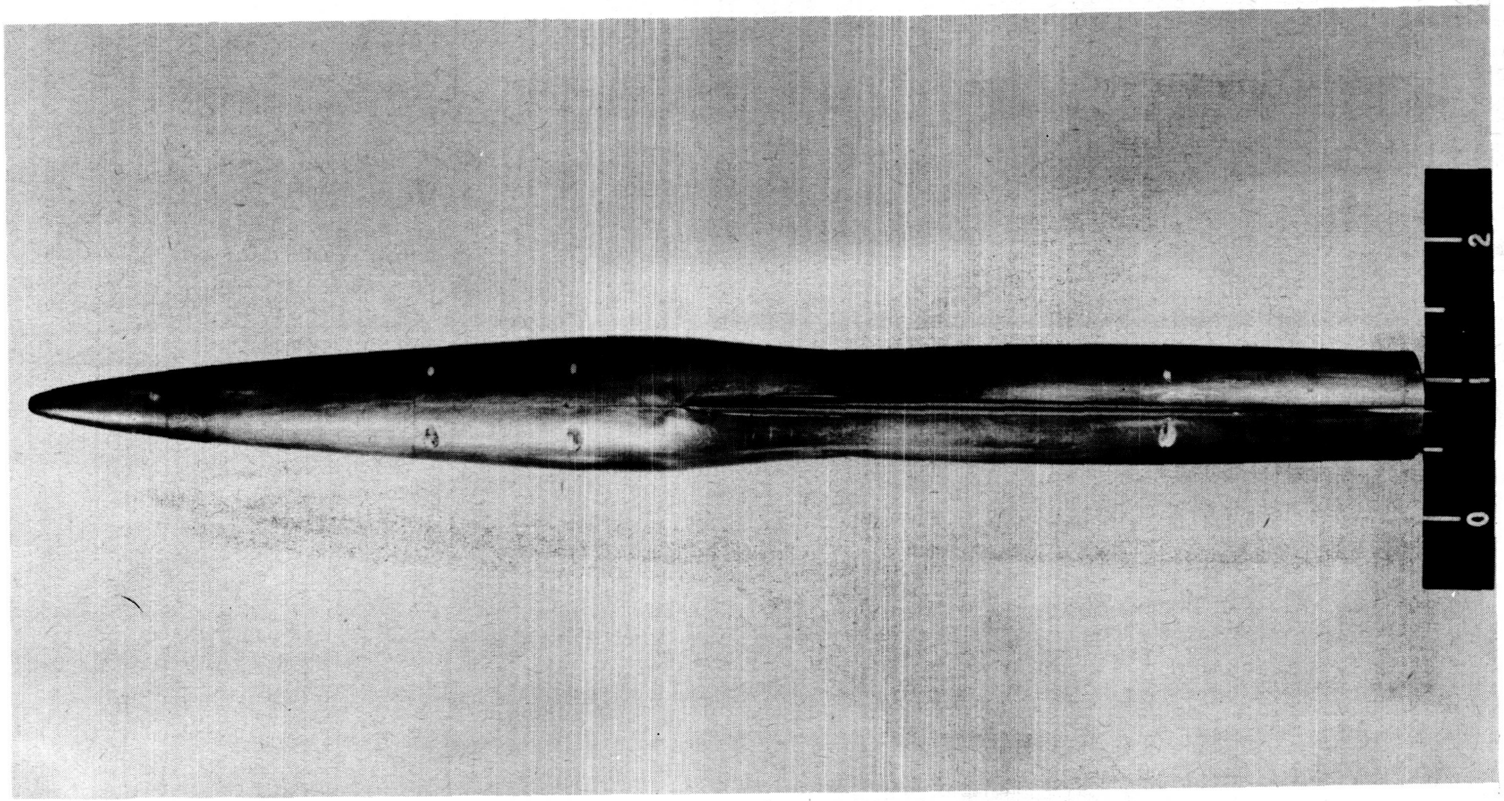
Wing data	
S, sq. in.	12.960
$\bar{c}$, in.	1.838
A	4.000

Figure 1.- Sketch showing the dimensions of the configurations tested. All dimensions are in inches unless otherwise noted.



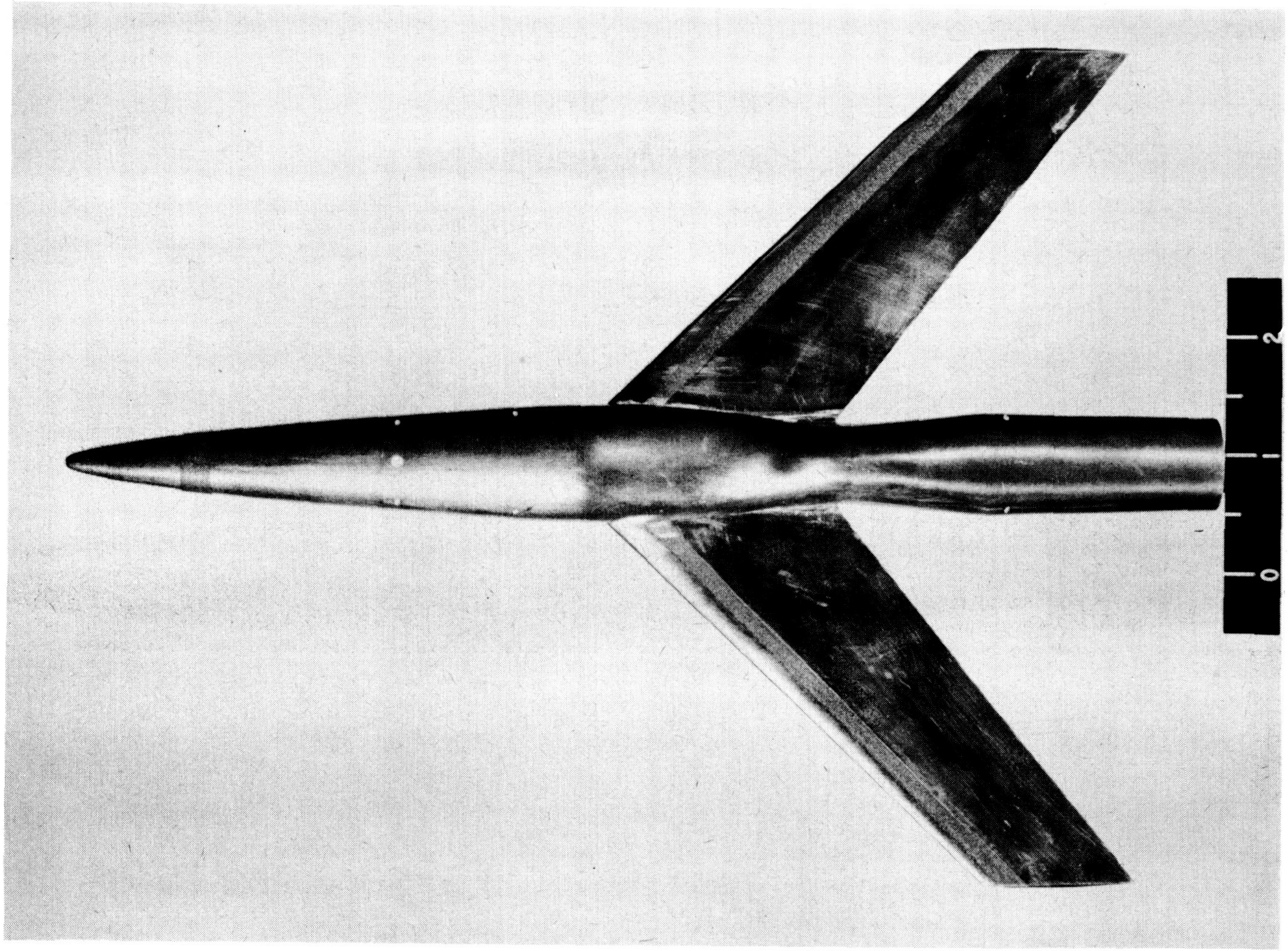
(a) Plan-form view. L-57-3589

Figure 2.- Plan-form and side-view photographs of the area-rule configuration.



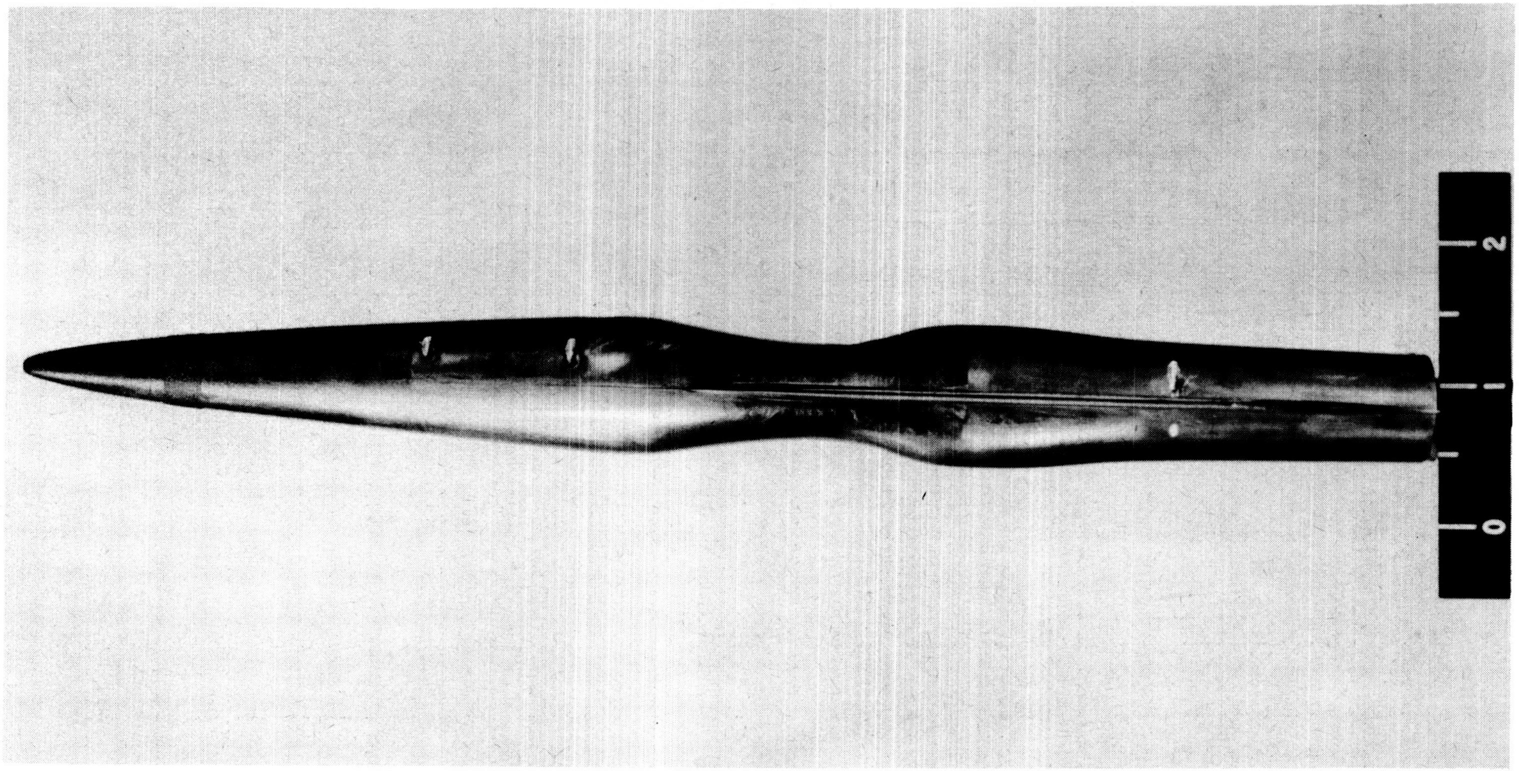
(b) Side view. L-57-3590

Figure 2.- Concluded.



(a) Plan-form view. L-57-3587

Figure 3.- Plan-form and side-view photographs of the streamline configuration.



(b) Side view. L-57-3588

Figure 3.- Concluded.

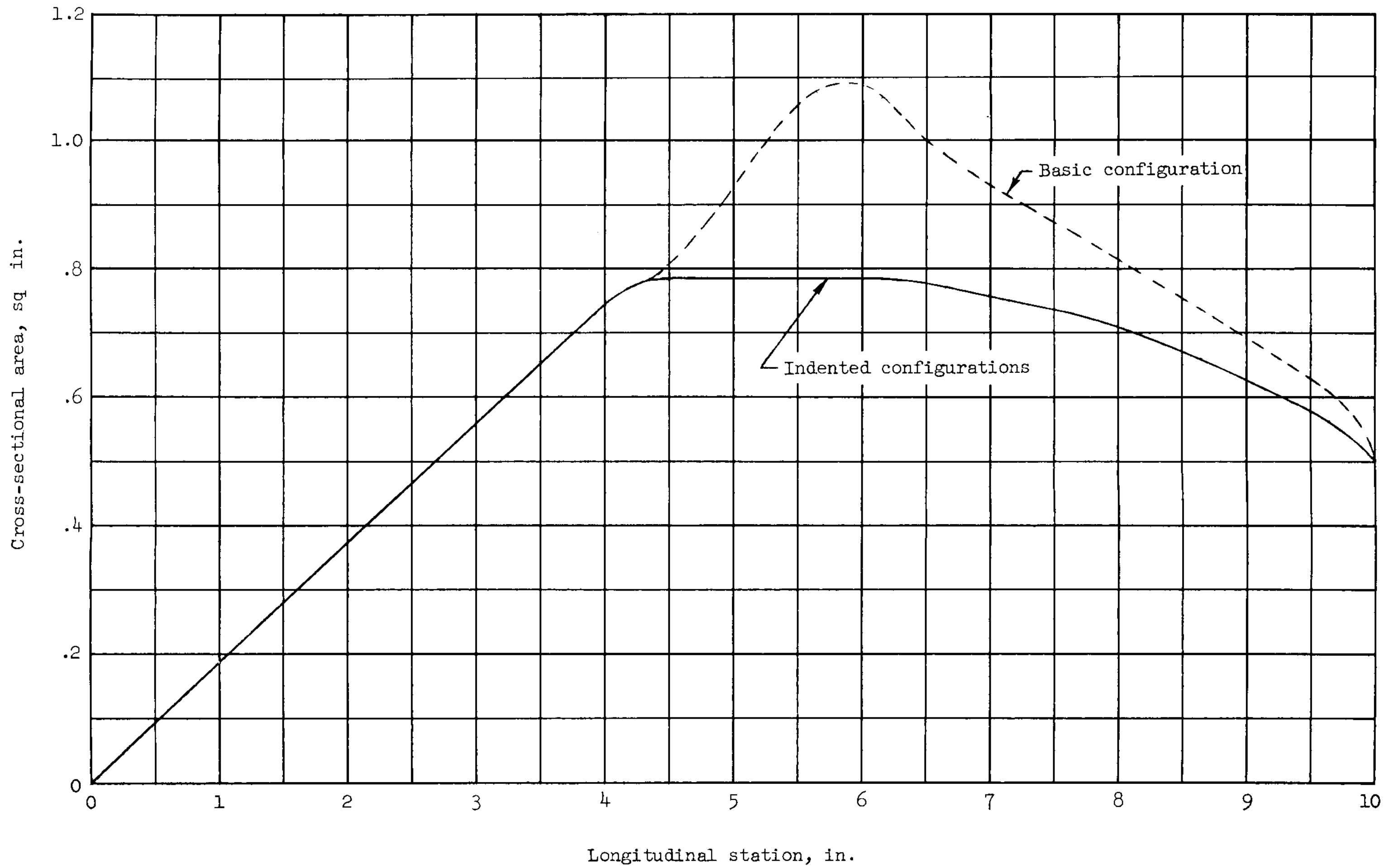


Figure 4.- Comparison of the longitudinal area distribution for the indented configurations with a configuration having no modification.

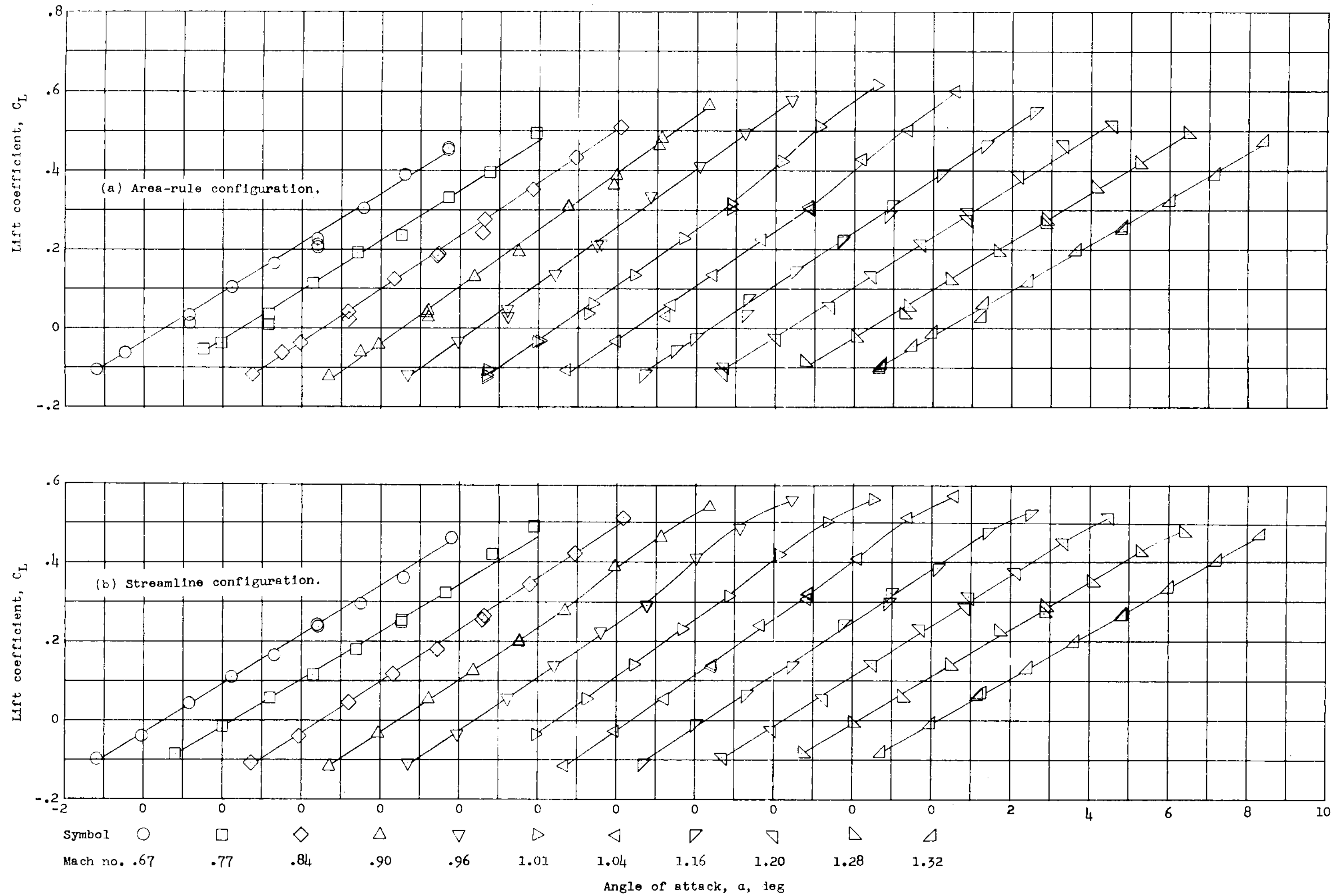
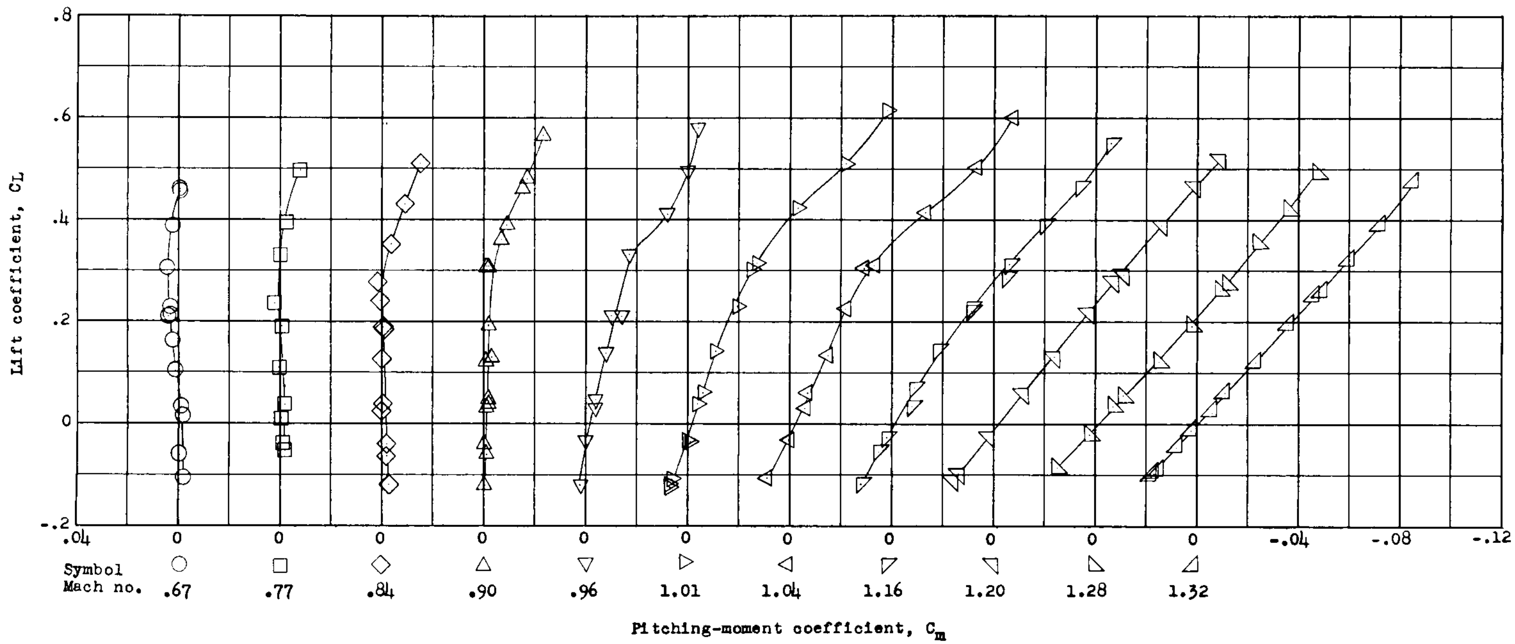
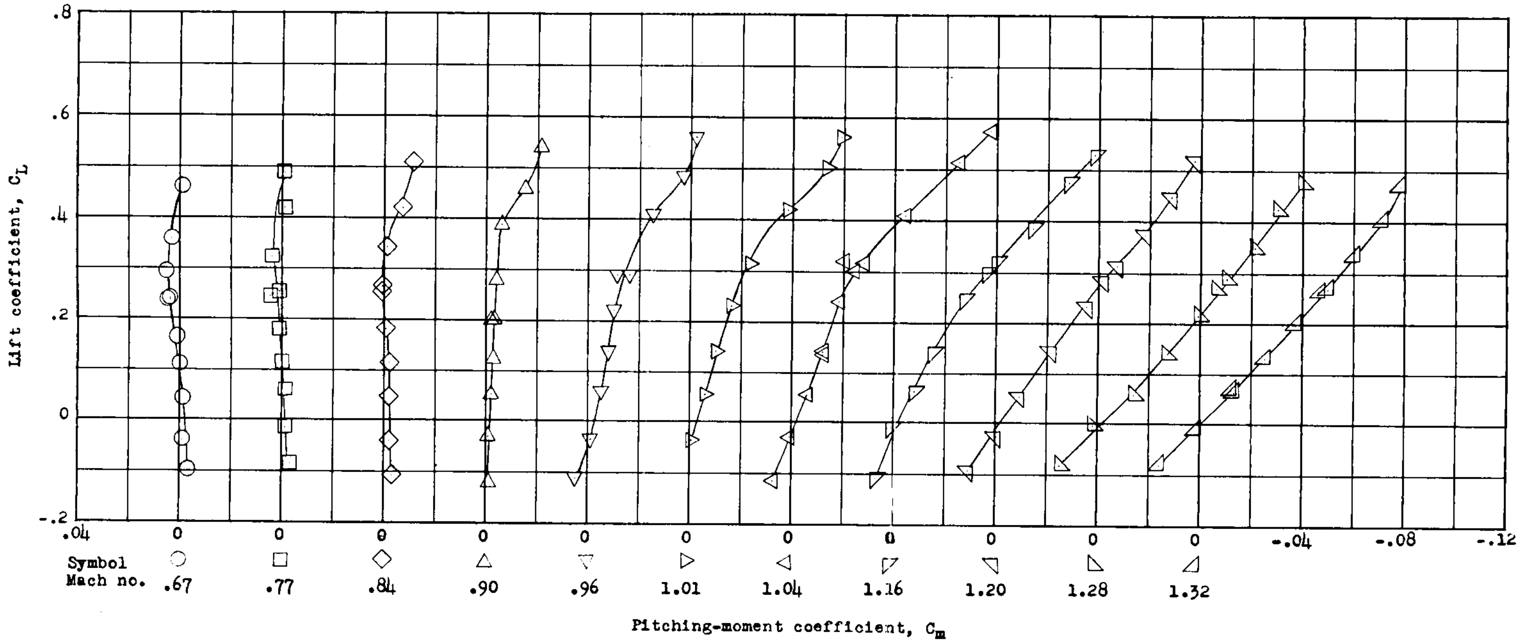


Figure 5.- Variation of lift coefficient with angle of attack at various Mach numbers.



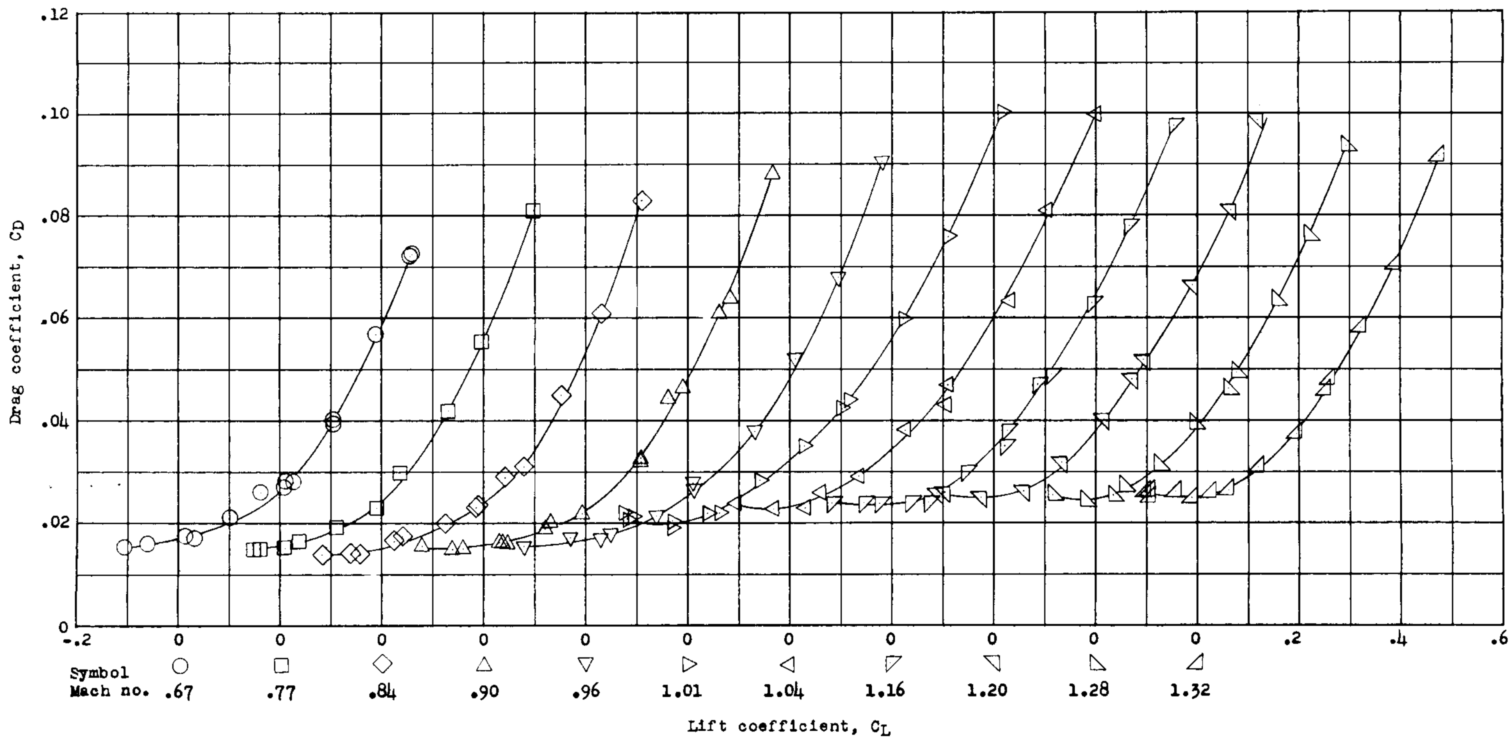
(a) Area-rule configuration.

Figure 6.- Variation of pitching-moment coefficient with lift coefficient at various Mach numbers.



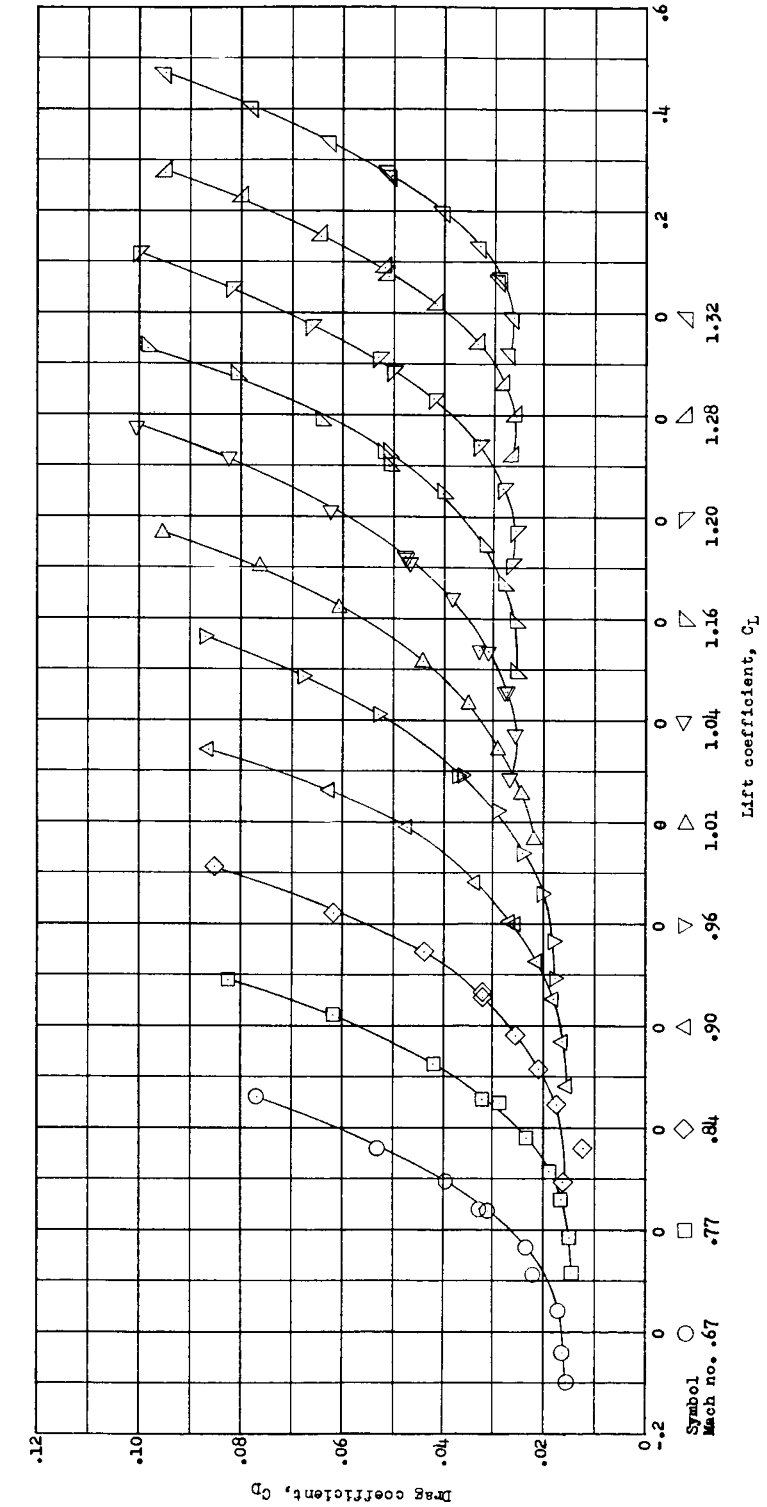
(b) Streamline configuration.

Figure 6.- Concluded.



(a) Area-rule configuration.

Figure 7.- Variation of drag coefficient with lift coefficient at various Mach numbers.



(b) Streamline configuration.

Figure 7.- Concluded.